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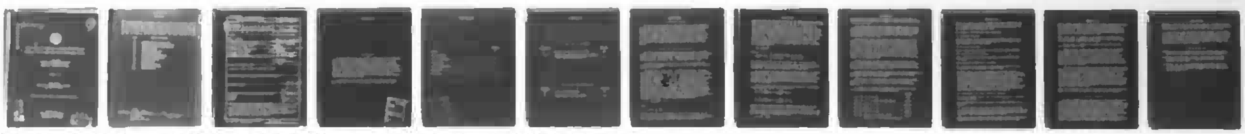

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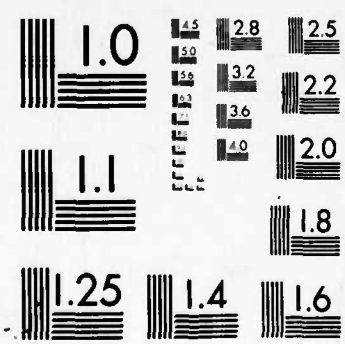

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USER'S GUIDE TO COMPUTER PROGRAM FOR ESTIMATING AIRPLANE LOCAL FLOW FIELD DUE TO WING-BODY EFFECTS

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1 APRIL 1977

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

This report describes a computer program, and its use, which has been written to determine, using a relatively simple model, the local flow field about an airplane due to wing-body effects. The program includes compressibility effects up to the critical Mach number. It takes as inputs various geometric parameters and the flight condition (in terms of Mach number, lift coefficient, and angle of attack).

S U M M A R Y

A computer program has been written to determine the local flow field about an airplane due to wing-body effects. The results are in terms of local angles of attack and sideslip at specified points. A simple geometric model is used with body effects determined from slender body theory, circulation effects (including wing-body interference) calculated using a series of line vortices, and wing thickness effects calculated using thin wing theory. The various geometric parameters plus the flight condition (in terms of Mach number, lift coefficient, and angle of attack) are input. (Span and chord loading may also be varied if desired.) Compressibility effects are accounted for but are limited to the subsonic (below critical) Mach number range.

A description of the program, including the assumptions made and model used, and of the use of the program is contained in this report.

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INTRODUCTION

For store separation studies or other purposes, it is often desirable to know the local flow field about an airplane (in terms of flow angularity). Wind tunnel tests, although best, are expensive and complicated and thus not practical in many cases. The alternative is some form of numerical calculation. Numerical computations, in attempts to become "exact", have a history of becoming very complicated without reproducing the actual flow, because assumptions and modeling still have to be made. In cognizance of all this, a computer program was developed using a relatively simple model for determining the local flow field about an airplane due to wing and body effects. The program, and the way to use it, are described in this report.

PROGRAM DESCRIPTION

A computer program (PROGRAM WBFLFD) has been written (for a CDC 6600 Computer) to obtain rough approximation of the local airplane flow field due to wing-body effects. The results are in terms of local angles of attack and sideslip at given (inputted) points. Compressibility effects are included but are limited to the subsonic (below critical) Mach number range.

The wing-body geometric model used is a straight-tapered wing with no dihedral and an axisymmetric body which may be at any height or incidence relative to one another.

The model used to generate the flow field is as follows: The local flow perturbations are considered to arise from three sources: circulation effects (mostly from the wing but including effects of wing-body interference), body effects, and wing thickness effects. The circulation effects are calculated by summing the flow velocities induced by a series of line vortices. The relative strengths of these vortices are determined such that desired spanwise and chordwise distributions are approximately reproduced (including the effects of wing-body interference). The absolute strengths of the vortices are determined such that a specified lift coefficient is produced. Figure 1 shows the arrangement of the line vortices on the wing and trailing behind the wing. Note that the theoretical wing (extending to body centerline) is used and that there are ten spanwise (in each semispan) divisions, between which vortex strength is varied and trailing vortices leave the wing, and three chordwise divisions. The spanwise lift distribution (in terms of circulation strength Γ at a given spanwise location y) is determined as follows: For an elliptical lift distribution,

$$\Gamma = \Gamma_0 \sqrt{1-(y/s)^2}$$

where

$$\Gamma_0 = \frac{4}{\pi} C_L \, c_{av} \left(\frac{U}{2} \right)$$

In these equations s is the semispan, C_L is the lift coefficient, c_{av} is the average chord, and U is the flight velocity. (For the present application, U is equal to 1.0 so the vertical and sideways perturbation velocity

components which are calculated are equal to the upwash and sidewash angles.) Wing-body interference effects cause a change in the spanwise lift distribution. This is modeled in the program as shown in Figure 2. The constants a and F shown there are determined such that the net area under the curve stays constant (that is, the same as in the elliptical distribution), which means the total lift stays the same, and also such that the ratio of exposed wing lift in the presence of the body to body lift due to the wing is in agreement with an accepted procedure for calculating this. (See, for example, reference (a).) The spanwise lift distribution can also vary due to such factors as twist, sweep, aspect ratio, taper ratio, and Mach number, which could cause either an inboard or an outboard shift in the distribution (without changing the total lift). Although usually the effect is small, this can be accounted for in the program by inputting a factor E such that at any spanwise location

$$\Delta\Gamma = \Delta(c_\ell c)\,\frac{U}{2}$$

where

$$\Delta(c_\ell c) = Ec_{av}\begin{cases} -3.82(y/s)^2 + 0.955, & \text{if } y/s \le 0.5 \\ \sin 2\pi(y/s), & \text{if } y/s > 0.5 \end{cases}$$

The variable c_ℓ is the local lift coefficient and c is the local chord. The values of the circulation at ten equally spaced spanwise locations (in each semispan) are used in the program calculations. (Some charts to estimate span loading changes are contained in reference (b).)

In the chordwise direction, the program nominally assumes a flat plate loading distribution, with line vortices located respectively at 0.135c, 0.25c, and 0.63c having relative magnitudes of 0.503, 0.345, and 0.152 (sum equals 1.0). The exact distribution is usually not critical, but may be varied in the program by adding increments to the first two relative magnitude values given above (with the third value automatically adjusted so that the sum remains 1.0).

Flow perturbations due to body effects are calculated in the program using slender body theory. The equations are:

$$\frac{w}{U} = R'(x)\,\frac{R(x)}{r}\sin\theta + \alpha_F\,\frac{R^2(x)}{r^2}\cos 2\theta$$

$$\frac{v}{U} = R'(x)\,\frac{R(x)}{r}\cos\theta - \alpha_F\,\frac{R^2(x)}{r^2}\sin 2\theta$$

w and v are the perturbation velocity components in the vertical (z) and sideways (y) directions respectively, R(x) is the body radius at the x-location of the inputted point in question, $R'(x)$ is the slope of the body at the x-location of the inputted point in question, r is the distance from the point in question to the body centerline,

$$\theta = \tan^{-1}\left(\frac{z-z_F}{y}\right)$$

where z_F is the z location of the fuselage (or body) centerline and y is measured from the airplane plane of symmetry, and α_F is the effective angle of attack of the body.

Flow perturbations due to wing thickness are generally small relative to the other contributions. Thus, the exact thickness distribution is not critical and an assumed thickness distribution (having a maximum thickness located at 0.368c) has been used in the program. The effects are assumed to be inversely proportional to the vertical distance from the wing, corresponding to thin wing theory.

Compressibility effects arise only in the calculation of velocities induced by the line vortices used to model the circulation effects. To correct for compressibility, all dimensions in the x direction are first multiplied by the factor $1/\sqrt{1-M^2}$ where M is the Mach number before the calculations for induced velocity are made. (The induced velocity, which is circumferential about the line vortex, would be inversely proportional to the distance from the line vortex for an infinitely long vortex; for a finite length or segment of a line vortex, though, it is somewhat less and depends on the distance and direction from the point to the ends of the vortex.) The vertical and sideways induced velocities then calculated are correct for the compressible flow.

The net local angle of attack at a given point is obtained as the sum of the actual angle of attack (relative to an airplane reference line) and the vertical perturbation velocities obtained from the three sources mentioned above (since U is set equal to 1.0 and the perturbation velocity in the x-direction is neglected relative to U).

The net local angle of sideslip is obtained as the sum of the perturbation velocities from the first two sources mentioned above (there not being any sidewash effect due to wing thickness).

Corrections to the calculated angles of attack and sideslip are made in the near vicinity of the wing and body to make the flow approach tangency to the wing or body surface as the distance to these surfaces diminishes.

The inputs to the program and corresponding formats are given below, with a description of the input parameters following:

		FORMAT
1st card:	NX, NY, NZ, NR	4I5
2nd card:	S, DM, B, C4SWP, CR, XCR4, TAU, ZF	8F10.4
3rd card:	X(I), I = 1, NX	8F10.4
4th card:	Y(J), J = 1, NY	8F10.4
5th card:	Z(K), K = 1, NZ	8F10.4
6th card:	AM, CL, AL, E, DELRM1, DELRM2	6F10.4
	(This last card to be repeated NR times.)	

NX, NY, NZ - number of x-values, y-values, and z-values respectively in a matrix of points (including all possible combinations) at each of which it is desired to calculate the local angles of attack and sideslip. NX, NY, and NZ must be in the range of 1 to 8.

NR - number of runs (different flight conditions).

S - theoretical wing area.

DM - body diameter at the location of the wing (should be equal to the horizontal distance between the exposed wing roots as is used for wing-body interference effect calculations).

B - theoretical wing span.

C4SWP - quarter chord sweep angle of the theoretical wing in degrees.

CR - theoretical wing root chord.

XCR4 - x-location of the quarter-chord point of the theoretical wing root chord.

TAU - thickness-to-chord ratio of the wing.

ZF - z-location of the fuselage (or body) centerline measured relative to the plane of the wing (positive, above the wing).

X(I), Y(J), Z(K) - x, y, and z-coordinates of the matrix of points at which local angles of attack and sideslip are to be calculated. X-values are referenced according to the value of XCR4 (more positive, farther aft), Y-values are referenced to the airplane vertical plane of symmetry (positive, out right wing), and Z-values are referenced to the plane of the wing (positive, above the wing). X(I), Y(J), and Z(K) should not be such that there are points in the plane of the wing (Z(K) = 0.) or on the fuselage centerline because of singularities in the mathematical model there. Also, Y(J) must be positive. (For desired points on the left side, input the corresponding point on the right side. The results, in terms of inboard or outboard sidewash, will be the same.)

AM - Mach number at the flight condition in question.

CL - lift coefficient, based on the theoretical wing area S, at the flight condition in question.

AL - airplane angle of attack (relative to an airplane reference line) in degrees.

E - a factor described above equal to the maximum $\Delta(c_\ell c)/c_{av}$ (occurring at the three-quarter semispan point) from an elliptical (neglecting wing-body interference effects) lift distribution at the flight condition in question (positive if lift is shifted from outboard to inboard).

NOTE: Effect of E is small so only need to input if there is a significant change from elliptical loading (not counting wing-body interference).

DELRM1, DELRM2 - increments to be added to the nominal relative magnitudes of the line vortex strengths of 0.503 for the line vortex positioned at 0.135c and of 0.345 for the line vortex positioned at 0.25c, respectively, as described above. (The sum of these two relative magnitudes plus that of the third line vortex located at 0.63c (set automatically) equals 1.0.) This defines the chordwise distribution of lift.

NOTE: Since effect of chordwise lift distribution is generally small, only need to input for DELRM1 and DELRM2 if there is quite a large change from the nominally assumed flat plate loading.

For all inputs of lengths or areas above, the same unit (or this unit squared) should be used.

Besides the input on data cards, certain variables must be defined with statements within the program (before they are used, obviously). RF(I), which corresponds to the body radius R(x) at the x-location corresponding to X(I), and RPF(I), which corresponds to $R'(x)$, must be defined for all NX values of I. (Most often, with the x-locations in question near the wing, RF(I) will be equal to 0.5 x DM; another note is that in the case of fuselages with oblong cross-sections, the horizontal dimension is usually what should be used (at least with regard to the flow field away from the fuselage).)

The effective fuselage angle of attack ALF and the wing angle of attack ALW, which can vary along the span, must also be defined. ALF is typically defined as the sum of AL plus any incidence of the fuselage relative to the airplane reference line (in degrees) minus an induced angle of attack ALI due to wing downwash. ALI is nominally set by the program based on the expected downwash at the wing for the given lift coefficient, area, and span. The wing angle of attack ALW, as presently set up in the program, is calculated as the sum of AL plus EO, the wing incidence relative to the airplane reference line, plus a wing twist factor ET (positive for washing) times the relative distance out the semispan. (EO and ET are in degrees and must be defined in the program.)

NOTE: ALW is used only to correct the flow field near the wing and does not affect other parts of the field; thus it is not critically important.

FINAL NOTES

It must be remembered, that even though it is done on a computer, the flow field calculated using this program is only a rough approximation. Factors affecting this are the fact that the geometry is simplified from the actual case, (usually, slender body and thin wing theory are used), the line vortices which are used to generate circulation effects are finite in number and the lift distribution (including wing-body interference effects) is only estimated; also, small angle approximations are used and the change in longitudinal velocity relative to the free-stream velocity is neglected in the calculations of downwash and sidewash angles. In addition, influences from other parts of the airplane, such as tail surfaces, are neglected. A typical error might be about 0.5 degrees (although in certain places it could be more), as indicated by comparison of the results in

Appendix B, which are for the A-7 airplane, with wind tunnel data reported in reference (c). The wind tunnel results themselves, at least as presented in that reference, are subject to errors of about the same magnitude.

It is possible, using the results from this program as a first order correction to free stream, to further refine the estimate of the flow field. As an example, the effect of a pod can be estimated by placing it in the flow field as determined by this program and then using slender body theory. Also, since the separate effects of the body, wing-body circulation, and wing thickness on the flow disturbance are indicated in the output, local corrections for any of these can be made.

REFERENCES

(a) Moore, Frank G., Aerodynamics of Guided and Unguided Weapons, Part I - Theory and Application, NWL Technical Report TR-3018, December 1973.

(b) DeYoung, John, and Harper, Charles W., Theoretical Symmetric Span Loading at Subsonic Speeds for Wings Having Arbitrary Plan Form, NACA Report No. 921, 1948.

(c) Carman, J. B., Jr., An Investigation of the Flow Field of the A-7D Aircraft with Several External Store Loadings at Mach Numbers 0.70 and 0.95, ARO, Inc., Report No. AD-A008476, Arnold Air Force Station, Tennessee, April 1975.

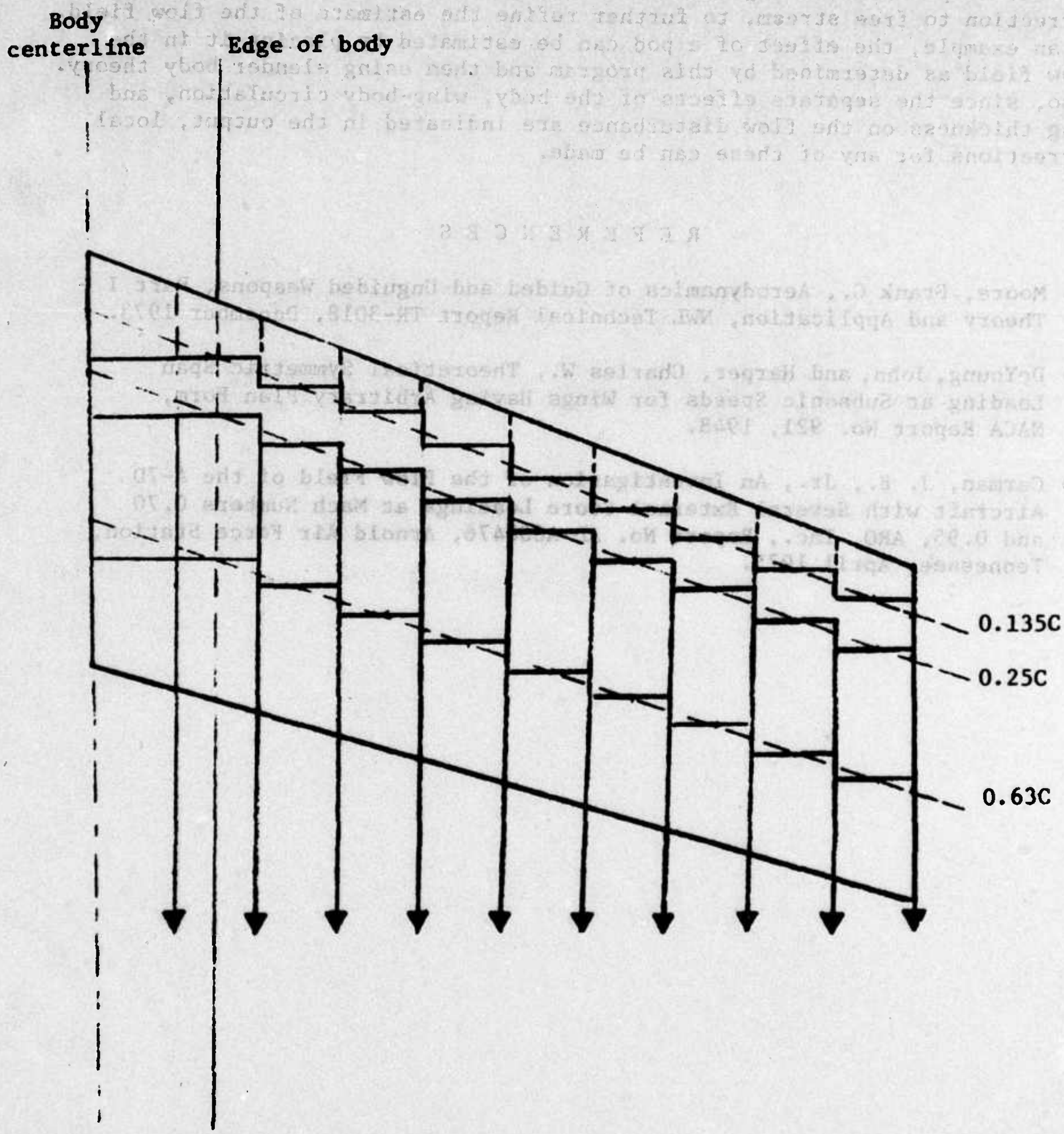
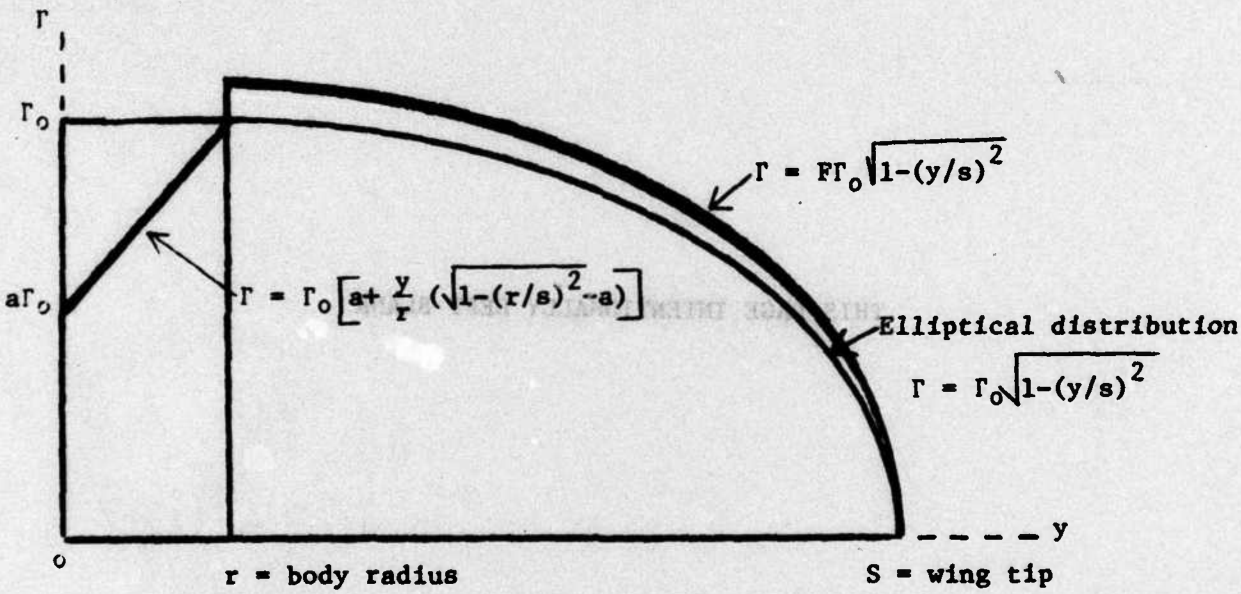


FIGURE 1. ARRANGEMENT OF LINE VORTICES ON WING



Heavy line indicates modified lift distribution to account for wing-body interference.

FIGURE 2. SPANWISE LIFT DISTRIBUTION CHANGE (AS MODELED) DUE TO WING-BODY INTERFERENCE

APPENDIX A

LISTING OF PROGRAM WBFLFD

APPENDIX A

```
      PROGRAM WBFLFD(INPUT,OUTPUT)
      DIMENSION X(8),Y(8),Z(8),RM(3),RF(8),RPF(8),YT(10),YG(10),XG(10,3)
     2,GM(10),GT(10)
      READ 5,NX,NY,NZ,NN
    5 FORMAT(4I5)
      READ 15,S,DM,B,C4SWP,CR,XCR4,TAU,ZF
   15 FORMAT(8F10.4)
      PRINT 20
   20 FORMAT(1H1,4X,*WING AREA*,3X,*BODY DIAM*,3X,*WING SPAN*,3X,
     2*C/4 SWEEP (DEG)   ROOT CHORD   X AT CR/4   WING THKNESS RATIO*,
     33X,*Z OF BODY CTRLINE*/)
      PRINT 25,S,DM,B,C4SWP,CR,XCR4,TAU,ZF
   25 FORMAT(3F12.2,F14.1,F17.2,F12.2,F16.2,F21.2)
      READ 15,(X(I),I=1,NX)
      READ 15,(Y(J),J=1,NY)
      READ 15,(Z(K),K=1,NZ)
      C=S/B
      CC=2.*(CR-C)
      RS=DM/B
      TLC4=TAN(C4SWP/57.296)
      TLLE=TLC4+.5/B*CC
      TLTE=TLC4-1.5/B*CC
      G=A=1.
      IF(TLTE.GT.0.) G=(CR-RS*CC)/(CR+.5*(B-DM)*TLC4-(.75+RS/4.)*CC)
      H=1.429*RS*G/(1.+.857*RS*G)
      F=1./(1.-RS*1.273*SQRT(1.-RS*RS/4.)*(1.+H))
      IF(RS.NE.0.) A=1.571/RS*H/(1.+H)-SQRT(1.-RS*RS)
      DO 40 I=1,NX
      RF(I)=.5*DM
   40 RPF(I)=0.
      EG=-1.
      ET=0.
      DO 50 M=1,10
      YT(M)=M*.05*B
      YG(M)=YT(M)-.025*B
      XG(M,1)=XCR4-.115*CR+YG(M)*(TLC4+.23/B*CC)
      XG(M,2)=XCR4+YG(M)*TLC4
   50 XG(M,3)=XCR4+.35*CR+YG(M)*(TLC4-.70/B*CC)
      DO 55 L=2,10
      DO 55 N=1,3
   55 IF(YG(11-L).LT.DM/2.) XG(11-L,N)=XG(12-L,N)
      DO 250 NN=1,NN
      READ 15,AM,CL,AL,E,DELRM1,DELRM2
      RM(1)=.503+DELRM1
      RM(2)=.345+DELRM2
      RM(3)=.152-DELRM1-DELRM2
      PRINT 60
   60 FORMAT(////,5X,*MACH NO.   LIFT COEFF.   ALPHA (DEG)   . E   *,30H
     2(E=MAX DELTA((CL*C)LOCAL)/C AV,9X,*RM(1)   RM(2)   RM(3)   (RM'S=REL.
     3 MAG. OF*/51X,*FROM ELLIPTICAL (POS., INBD SHIFT)*,26X,*CHORDWISE LI
     4FT DIST.*/111X,*AT .135C, .25C, .63C)*)
      PRINT 65,AM,CL,AL,E,RM(1),RM(2),RM(3)
   65 FORMAT(1H0,F9.2,F13.3,F15.2,F11.3,37X,3F7.3)
      BETA=SQRT(1.-AM*AM)
      CL=CL-.02*DM*DM/S*AL
      G0=.5*B*CL*C
      ALI=20.*CL*S/B/B
      ALF=AL-ALI
      DO 70 M=1,10
      YS=YG(M)/B*2.
      IF(YS.GE.RS) GM(M)=F*G0*SQRT(1.-YS*YS)
      IF(YS.LT.RS) GM(M)=G0*(A+(SQRT(1.-RS*RS)-A)*YS/RS)
      IF(E.NE.0..AND.YS.GT..5) GM(M)=GM(M)+.5*F*C*SIN(6.283*YS)
```

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70    IF(F.NF.0..AND.YS.LT..5) GN(M)=GN(M)*[illegible]
      GT(10)=GN(10)
      DO 80 M=1,9
80    GT(M)=GN(M)-GN(M+1)
      PRINT 90
90    FORMAT(///,7X,*Y*,8X,*X*,8X,*Z*,5X,*ALPHA LOCAL (DEG)    BETA LOCAL
     2 (DEG)    ALPHA DUE     BETA DUE    ALPHA DUE     BETA DUE    ALPHA DU
     3E TO*/33X,*(POSITIVE UP)      (POS., OUTBOARD)    TO W-B CIRC   TO W-
     4B CIRC   TO BODY        TO BODY      WING THKNESS*/)
      DO 250 J=1,NY
      CH=CR+Y(J)*(TLTE-TLLE)
      ALW=AL+F0+ET*2.*Y(J)/B
      DO 250 I=1,NX
      DO 250 K=1,NZ
      WW=VW=WTH=WF=VF=0.
      DO 100 M=1,10
      YY=Y(J)-YT(M)
      YYS=Y(J)+YT(M)
      DO 100 N=1,3
      XX=(X(I)-XG(M,N))/BTA
      IF(M.LE.9) XX1=(X(I)-XG(M+1,N))/BTA
      IF(M.EQ.10) XX1=XX
      D=SQRT(XX*XX+YY*YY+Z(K)*Z(K))
      DS=SQRT(XX*XX+YYS*YYS+Z(K)*Z(K))
      D1=SQRT(XX1*XX1+YY*YY+Z(K)*Z(K))
      DS1=SQRT(XX1*XX1+YYS*YYS+Z(K)*Z(K))
      FF=GN(M)*HM(M)*4.559
      FC=-FF*XX/(XX*XX+Z(K)*Z(K))
      WW=WW+FC*(YYS/DS-YY/D)
      IF(M.GT.1) WW=WW+FC*((Y(J)-YT(M-1))/SQRT(XX*XX+(Y(J)-YT(M-1))*
     2(Y(J)-YT(M-1))+Z(K)*Z(K))-(Y(J)+YT(M-1))/SQRT(XX*XX+(Y(J)+YT(M-1))
     3*(Y(J)+YT(M-1))+Z(K)*Z(K)))
      R2=YY*YY+Z(K)*Z(K)
      RS2=YYS*YYS+Z(K)*Z(K)
      IF(M.LE.9) WW=WW+FF*(YY/R2*(XX/D-XX1/D1)-YYS/RS2*(XX/DS-XX1/DS1))
      IF(M.LE.9) VW=VW+FF*Z(K)*((XX/DS-XX1/DS1)/RS2-(XX/D-XX1/D1)/R2)
      IF(N.NE.2) GO TO 100
      WW=WW+GT(M)*4.559*(YY/R2*(XX1/D1+1.)-YYS/RS2*(XX1/DS1+1.))
      VW=VW+GT(M)*4.559*Z(K)*((XX1/DS1+1.)/RS2-(XX1/D1+1.)/R2)
100   CONTINUE
      IF(RF(I).LE.0.) GO TO 120
      ZZF=Z(K)-ZF
      R2=Y(J)*Y(J)+ZZF*ZZF
      WF=(RPF(I)*ZZF*57.296+ALF*RF(I)*(Y(J)*Y(J)-ZZF*ZZF)/R2)*RF(I)/R2
      VF=(RPF(I)*Y(J)*57.296-ALF*RF(I)*2.*ZZF*Y(J)/R2)*RF(I)/R2
120   W=WW+WF
      IF(Y(J).GE.B/2..OR.Y(J).LT.DF/2.) GO TO 200
      XC=(X(I)-XCR-.CH/4.-Y(J)*TLLE)/CH
      IF(XC.LE.0..OR.XC.GT.1.) GO TO 200
      IF(ABS(Z(K)).LT..25*CH/BTA) W=-ALW+4.*ABS(Z(K))/CH*BTA*(W+ALW)
      WTH=(AU*TAU/Z(K)*CH*(((17.83*XC-60.84)*XC+89.66)*XC-50.13)*XC+
     29.28)*XC*57.296
      W=WW+WTH+WF
200   IF(RF(I).LE.0.) GO TO 210
      RHF=SQRT(R2)/RF(I)
      IF(RHF.LT.1.5.AND.Y(J)*Y(J)/R2.LT..5) WW=-AL+2.*(RHF-1.)*(WW+AL)
      IF(RHF.LT.1.5.AND.Y(J)*Y(J)/R2.GT..5) VW=2.0*VW*(RHF-1.)
210   W=WW+WF+AL
      V=VW+VF
      W=W-WW-WTH
250   PRINT 260,Y(J),X(I),Z(K),W,V,WW,VW,WF,VF,WTH
260   FORMAT(3X,3F9.2,F14.2,F10.2,5X,5F12.2)
      STOP
      END
```

APPENDIX B

SAMPLE OUTPUT FROM PROGRAM WBFLFD

APPENDIX B

WING AREA	BODY DIAM	WING SPAN	C/4 SWEEP (DEG)	ROOT CHORD	X AT C/4	WING THKNESS RATIO	Z OF BODY CTRLINE
375.00	6.00	30.75	45.0	15.49	32.31	.07	-2.56

MACH NO.	LIFT COEFF.	ALPHA (DEG)	F	(EMAX DELTA((CL*C)LOCAL)/C AV FROM ELLIPTICAL (POS.. INBD SHIFT)	RM(1)	RM(2)	RM(3)	(UNTSPHEL. MAG. OF CHORDWSF LIFT DIST. AT .135C, .25C, .43C)
.80	.300	5.42	-.020		.503	.345	.152	

Y	X	Z	ALPHA LOCAL (DEG) (POSITIVE UP)	BETA LOCAL (DEG) (POS.. OUTBOARD)	ALPHA DUE TO V-D CIRC	BETA DUE TO V-D CIRC	ALPHA DUE TO BODY	BETA DUE TO BODY	ALPHA DUE TO WING THKNESS
3.00	30.00	-3.00	4.49	.93	1.05	.10	2.65	.79	.00
3.00	30.00	-3.70	4.45	1.62	1.00	.10	1.04	1.04	.00
3.00	30.00	-7.00	5.44	1.03	.32	.21	-.33	.02	.00
3.00	30.00	-13.00	5.77	.22	-.05	.11	-.18	.11	.00
3.00	30.00	-20.00	5.00	.08	-.12	.06	-.08	.03	.00
3.00	33.44	-3.00	5.09	1.43	-2.91	.63	2.65	.79	-.05
3.00	33.44	-3.70	5.79	2.26	-1.53	.63	1.04	1.04	-.52
3.00	33.44	-7.00	4.92	1.26	-.47	.42	-.33	.02	-.29
3.00	33.44	-13.00	5.30	.26	-.36	.17	-.18	.11	-.18
3.00	33.44	-20.00	5.50	.11	-.27	.04	-.08	.03	-.18
3.00	37.74	-3.00	3.70	.46	-5.21	-.33	2.65	.79	.35
3.00	37.74	-3.70	4.28	1.52	-3.92	-.11	1.94	1.64	.28
3.00	37.74	-7.00	3.40	1.26	-1.04	.30	-.33	.02	.15
3.00	37.74	-13.00	5.01	.33	-.09	.72	-.18	.11	.08
3.00	37.74	-20.00	4.48	.13	-.50	.19	-.08	.03	.05
3.00	43.34	-3.00	3.62	.67	-5.12	-.73	2.65	.79	.09
3.00	43.34	-3.70	3.90	1.07	-4.02	-.57	1.04	1.04	.07
3.00	43.34	-7.00	3.24	.96	-2.47	.14	-.33	.02	.04
3.00	43.34	-13.00	4.45	.34	-1.39	.23	-.18	.11	.02
3.00	43.34	-20.00	5.17	.16	-.76	.13	-.08	.03	.01
3.00	47.00	-3.00	4.18	.81	-3.49	-.00	2.65	.79	.00
3.00	47.00	-3.70	4.70	.57	-3.14	-.07	1.04	1.64	.00
3.00	47.00	-7.00	3.25	.87	-2.02	.05	-.33	.02	.00
3.00	47.00	-13.00	4.29	.24	-1.52	.22	-.14	.11	.00
3.00	47.00	-20.00	5.65	.17	-.04	.14	-.08	.03	.00
5.10	30.00	-3.00	4.10	.53	1.23	.37	.96	.17	.00
5.10	30.00	-3.70	7.40	.76	1.02	.34	.04	.40	.00
5.10	30.00	-7.00	6.05	.06	.37	.31	.08	.55	.00
5.10	30.00	-13.00	5.07	.32	-.02	.17	-.12	.15	.00
5.10	30.00	-20.00	5.82	.13	-.11	.09	-.06	.04	.00
5.10	33.44	-3.00	5.05	2.21	-.04	2.05	.96	.17	-.43
5.10	33.44	-3.70	5.45	1.94	-.05	1.54	.04	.40	-.25
5.10	33.44	-7.00	5.05	1.23	-.24	.68	.08	.55	-.18
5.10	33.44	-13.00	5.40	.42	-.31	.27	-.12	.15	-.10
5.10	33.44	-20.00	5.62	.17	-.25	.13	-.06	.04	-.06
5.10	37.74	-3.00	2.14	1.77	-4.07	1.60	.96	.17	.10
5.10	37.74	-3.70	3.16	1.80	-3.76	1.40	.04	.40	.08
5.10	37.74	-7.00	4.41	1.37	-1.71	.02	.08	.55	.04
5.10	37.74	-13.00	5.09	.52	-.91	.37	-.12	.15	.02
5.10	37.74	-20.00	5.40	.21	-.47	.17	-.06	.04	.01
5.10	43.34	-3.00	1.90	.36	-5.20	.20	.96	.17	.15
5.10	43.34	-3.70	2.51	.66	-4.43	.25	.94	.40	.12
5.10	43.34	-7.00	3.52	1.06	-2.92	.51	.08	.55	.04
5.10	43.34	-13.00	4.54	.55	-1.34	.40	-.12	.15	.03
5.10	43.34	-20.00	5.23	.26	-.73	.21	-.06	.04	.02
5.10	47.00	-3.00	2.54	.07	-4.41	-.04	.96	.17	.00
5.10	47.00	-3.70	2.64	.39	-3.95	-.01	.44	.40	.00

5.10	47.00	-7.00	3.65	.42	-2.03	.37	.08	.55	.00
5.10	47.00	-13.00	4.34	.54	-1.49	.49	-.12	.15	.00
5.10	47.00	-20.00	5.04	.27	-.05	.23	-.06	.04	.00
8.10	30.00	-3.00	7.47	.37	1.04	.32	.34	.04	.00
8.10	30.00	-3.70	7.32	.46	.45	.36	.36	.10	.00
8.10	30.00	-7.00	6.60	.62	.44	.36	.16	.25	.00
8.10	30.00	-13.00	6.00	.36	.03	.23	-.06	.14	.00
8.10	30.00	-20.00	5.67	.15	-.08	.14	-.06	.05	.00
8.10	33.44	-3.00	4.05	1.39	1.00	1.35	.38	.04	.00
8.10	33.44	-3.70	7.55	1.34	1.20	1.29	.38	.10	.00
8.10	33.44	-7.00	6.36	1.06	.15	.61	.15	.25	.00
8.10	33.44	-13.00	5.77	.51	-.14	.37	-.06	.14	.00
8.10	33.44	-20.00	5.75	.24	-.41	.17	-.06	.05	.00
8.10	37.74	-3.00	3.64	3.71	-2.28	3.67	.38	.04	-.41
8.10	37.74	-3.70	4.34	3.05	-1.64	2.96	.36	.10	-.33
8.10	37.74	-7.00	4.40	1.54	-1.09	1.34	.16	.25	-.17
8.10	37.74	-13.00	5.23	.66	-.64	.54	-.06	.14	-.09
8.10	37.74	-20.00	5.45	.31	-.41	.25	-.06	.05	-.05
8.10	43.30	-3.00	1.66	1.92	-4.76	1.67	.38	.04	.24
8.10	43.30	-3.70	2.60	1.66	-4.16	1.73	.36	.10	.19
8.10	43.34	-7.00	3.81	1.45	-2.46	1.20	.16	.25	.10
8.10	43.34	-13.00	4.63	.76	-1.19	.64	-.06	.14	.06
8.10	43.34	-20.00	5.36	.36	-.66	.33	-.06	.05	.04
8.10	47.00	-3.00	2.12	1.38	-4.20	1.36	.38	.04	.00
8.10	47.00	-3.70	2.43	1.36	-3.93	1.27	.36	.10	.00
8.10	47.00	-7.00	3.56	1.20	-2.62	1.03	.16	.25	.00
8.10	47.00	-13.00	4.59	.79	-1.38	.65	-.06	.14	.00
8.10	47.00	-20.00	5.19	.41	-.77	.35	-.04	.05	.00
11.00	30.00	-3.00	7.67	.25	.00	.24	.19	.02	.00
11.00	30.00	-3.70	7.00	.31	.01	.27	.19	.04	.00
11.00	30.00	-7.00	6.54	.45	.47	.34	.13	.11	.00
11.00	30.00	-13.00	6.18	.37	.09	.27	.01	.11	.00
11.00	30.00	-20.00	5.93	.22	-.05	.17	-.02	.05	.00
11.00	33.44	-3.00	7.63	.72	1.44	.70	.19	.02	.00
11.00	33.44	-3.70	7.41	.88	1.22	.76	.19	.04	.00
11.00	33.44	-7.00	6.55	.63	.43	.72	.13	.11	.00
11.00	33.44	-13.00	5.95	.53	-.06	.42	.01	.11	.00
11.00	33.44	-20.00	5.63	.26	-.15	.23	-.02	.05	.00
11.00	37.74	-3.00	7.65	2.88	1.46	2.66	.19	.02	.00
11.00	37.74	-3.70	7.02	2.56	.43	2.53	.19	.04	.00
11.00	37.74	-7.00	5.69	1.55	-.23	1.44	.13	.11	.00
11.00	37.74	-13.00	5.61	.75	-.40	.65	.01	.11	.00
11.00	37.74	-20.00	5.66	.38	-.32	.32	-.02	.05	.00
11.00	43.30	-3.00	2.64	3.23	-3.67	3.22	.19	.02	.24
11.00	43.30	-3.70	3.23	2.45	-3.18	2.91	.19	.04	.23
11.00	43.30	-7.00	4.43	1.91	-1.82	1.79	.13	.11	.12
11.00	43.30	-13.00	5.15	.95	-.92	.85	.01	.11	.06
11.00	43.30	-20.00	5.45	.47	-.54	.42	-.02	.05	.04
11.00	47.00	-3.00	2.35	2.50	-3.44	2.48	.19	.02	.00
11.00	47.00	-3.70	2.71	2.39	-3.08	2.35	.19	.04	.00
11.00	47.00	-7.00	3.92	1.81	-2.21	1.70	.13	.11	.00
11.00	47.00	-13.00	4.88	1.01	-1.13	.90	.01	.11	.00
11.00	47.00	-20.00	5.33	.52	-.05	.46	-.02	.05	.00

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